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## ELECTROTHERMAL MICROTHRUST SYSTEMS

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### Abstract

A limited treatment of design and performance characteristics of several electrothermal micro-thrust systems is presented. Two systems described have been incorporated into spacecraft. The spacecraft functions for which they are being used or contemplated for use are attitude control, station keeping, and station seeking. Heated ammonia fast heat-up systems are proposed for station keeping of the synchronous orbit, gravity gradient stabilized spacecraft (D and E) in the Applications Technology Satellite series. Experimental versions being flown earlier are described. Single jet and multijet thermal storage thrusters for systems application are currently in an advanced stage of development. Design configuration and curves of specific impulse and temperature versus power are shown. These units have thrust levels ranging from five millipounds to fifty millipounds at specific impulses greater than two hundred and forty seconds. The multijet thrusters incorporate as many as five exit nozzles. Tables of performance data show current development status of these units.

### Introduction

The electrothermal propulsion system is now a serious contender for application to spacecraft for attitude control, station keeping, and station seeking. The transition from the laboratory into spacecraft projects, however, is introducing a new category of problems related to integration with the spacecraft and accommodating its mission objectives. Additional effort directed toward the solution of these problems is required. Substantial progress has been made in hardware development and it is evident from the present status that acceptance of electrothermal systems as advanced, efficient propulsion systems suitable for application is becoming well established.

Systems are being flown as prime mission hardware and as experiments. Other hardware items are in advanced stages of development. TRW Systems, under contract to the Air Force, has developed a nitrogen gas electrothermal system for the Vela Advanced Spacecraft. Avco, under contract to Goddard Space Flight Center, built and integrated a fast heat-up ammonia resistojet system as an experiment aboard an Applications Technology Satellite (ATS-1). A follow-on experiment is to be flown aboard the ATS-C spacecraft; the design is being proposed for synchronous orbit station keeping of the gravity gradient oriented ATS-D and ATS-E spacecraft. General Electric, under contract to Goddard Space Flight Center, is in an advanced stage of development of single jet and multijet thermal storage thrusters. Marquardt, under sponsorship of NASA, is developing a tubular concentric resistojet.

Other systems are being flown or contemplated for flight but because of security classification or proprietary interests cannot be discussed herein. This paper will attempt to provide a limited discussion of the functions, description, and performance of the examples mentioned above.

### Vela Advanced Spacecraft Electrothermal System\*

#### Function

The electrothermal system is a nitrogen gas system designed to control the velocity, spin rate, and orientation of the Vela Advanced Spacecraft in orbit. The system and spacecraft configuration are depicted in Figure 1. The thruster contains three nozzles through which the gaseous propellant is expanded to provide thrust when demanded by ground command or control electronics. A resistance heater is incorporated in the thruster design. When electric power is available from the spacecraft power supply, the heater may be used to improve performance by heating the propellant prior to expansion through the nozzle.

Two Vela Advanced Spacecraft will be launched in tandem aboard a Titan III-C booster. Their injection into a circular orbit will be accomplished in the manner of the Vela 3. The Vela spacecraft are launched in pairs into an elliptical orbit of approximately 104 mile perigee and 60,000 mile apogee. When the pair reach apogee, a solid rocket in one of the spacecraft is fired, injecting that spacecraft into a circular orbit. The remaining spacecraft is allowed to complete an additional elliptical orbit and is then injected into a circular orbit at its second apogee. Timing is such that a spacing of 140 degrees is obtained between the two spacecraft. Once in circular orbit, the spacecraft is despun and attitude oriented by the electrically heated thrusters. When the relative orbital position of the two spacecraft has been determined, velocity corrections are performed by the heated thrusters. Because power is obtained from batteries, it is not continuously available. Thus, the sustaining attitude control propulsion is generally performed with the thrusters unheated. Total impulse requirement for all maneuvers is approximately 1300 lb-sec. With leakage and ullage allowances, 14.2 lbs. of propellant are carried on each spacecraft.

#### Description

As shown in Figure 1, two thruster assemblies are located 180° apart on the perimeter of the spacecraft in the plane which passes through the center of gravity perpendicular to the spin axis. Each

\*Courtesy of TRW Systems

thruster has three nozzles located 2.3 feet from the spin axes. Each nozzle is connected to a valve which allows propellant to flow on command.

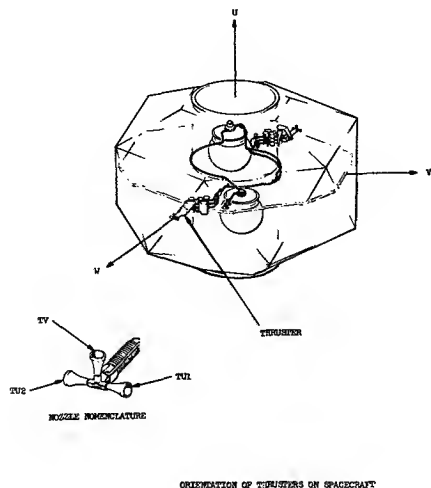


FIGURE 1 - VELA SPACECRAFT/THRUSTER CONFIGURATION

The orientation of the spacecraft is controlled by the TU nozzles. They are fired individually when required by control logic. Spin control is performed by simultaneously firing two TU nozzles which point in opposite directions parallel to the V axis, one on each thruster. A maneuver will usually consist of a continuous firing which is terminated when the spacecraft is spinning at a desired rate. A velocity correction is made by firing two of the TU nozzles which point in the same direction parallel to the V axis. The gas flow is pulsed so that thrust is developed only when the spacecraft V axis is pointing in the proper direction parallel to the flight path. To perform velocity corrections in the shortest possible time, opposing pairs of TU nozzles can be fired alternately.

Propellant is stored at 4000 psia in two symmetrically located tanks. A single stage regulator with an integral relief valve on the low pressure side is used to maintain thruster inlet pressure at a nominal 50 psia. Both high and low pressure transducers are used to monitor system operation. Six valves are used to control the six thrust nozzles. The total system weight, exclusive of propellant, is 23.1 pounds.

A basic thruster unit, shown in figure 2, has three nozzles, each fed by an individual propellant flow tube. Two propellant flow tubes, which supply propellant to the two TU nozzles, are coiled around the core assembly. The coiled tubes serve as heat exchangers for raising the propellant temperature during hot duty cycle operation since the heater element is also wound on the core.

Figure 3 shows a completely assembled thruster module. A case assembly that is formed in two halves surrounds the basic thruster unit. The case is mounted on a thruster support bracket; it serves as an extra support member for the thruster during prelaunch and launch period and serves as a radiation shield during hot cycle operation. To minimize the thermal radiation heat losses from the

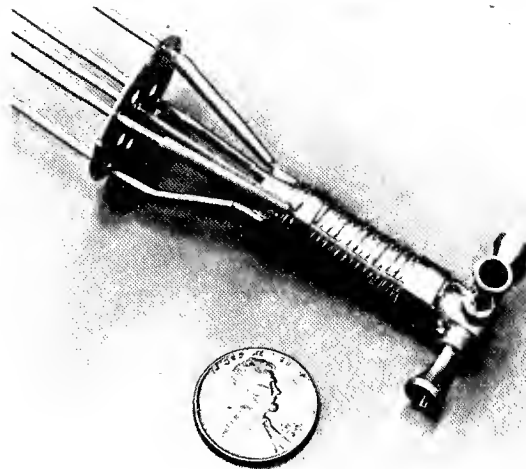


FIGURE 2 - TRW UNSHIELDED THRUSTER UNIT

thruster, the heater assembly and case assembly have an outer surface layer of gold. A unique feature is the use of three Delrin buttons to form a structural bridge between the heater and case assembly. The buttons are mounted in set screws that are threaded in the case assembly and contact the heater assembly. The Delrin buttons which are preloaded, sublime during the initial thruster pre-heat after launch. They are no longer required for structural support after launch and, by subliming, eliminate a thermal loss path during the hot cycle operations.

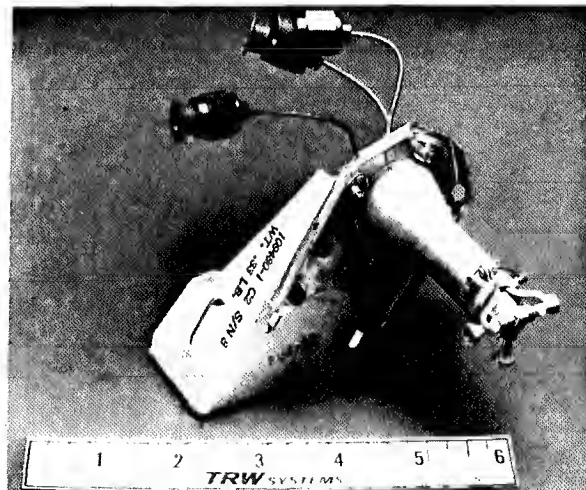


FIGURE 3 - TRW SHIELDED THRUSTER UNIT

#### Performance

Table 1 provides the primary nominal operating parameters of a thruster system.

TABLE 1

1. Propellant; gaseous nitrogen with 2% argon by volume
2. Power; 28 to 35 watts (steady propellant flow)  
15 to 20 watts (pulsed propellant flow)  
16 to 19 watts (no flow)
3. Voltage;  $20.5 \pm 1.5$  volts (nominal battery bus)
4. Gas Temperature  $550 \pm 100^\circ\text{F}$  (steady propellant flow)

- 1250  $\pm$  100°F (pulsed propellant flow)
5. Maximum normal heater temperature; 1430°F (no propellant flow)
  6. Power loss; 9 watts nominal at 1000°F
  7. Flow rate 2 x 10<sup>-4</sup> lb/sec (steady propellant flow)  
1.5 x 10<sup>-4</sup> lb/sec (propellant flow during a pulse)
  8. Heater inlet gas pressure; 49  $\pm$  3 psia
  9. Duty Cycle;  
Hot gas: despin - 106 hrs. continuous  
 $\Delta V$  - 3 seconds on/29 seconds off  
Ambient gas: control - 2 milliseconds on/  
800 seconds off
  10. Specific Impulse;  
Hot gas: 96 - 100 seconds (steady propellant flow)  
130 - 135 seconds (pulsed propellant flow)  
Ambient gas: 70 seconds nominal
  11. Nozzle area ratio; 67:1
  12. Thrust; .019 lbs. nominal
  13. Total impulse;  
Hot gas: 739 lb-sec  
Ambient: 502 lb-sec
  14. Thruster weight; .33 lb.

Following essentially the technology upon which the Vela system was designed, TRW Systems is currently performing advanced development in low pressure liquid ammonia propellant electrothermal systems which advancements should result in performance superior to that of the system described.

#### Flight Experiments

Liquid ammonia propellant electrothermal systems are being given serious consideration by principal investigators of advanced systems. Operating pressures are relatively low and the basic physical properties of ammonia provide specific impulses which enhance system propulsion efficiency. Moreover, ammonia systems are relatively safe to integrate into a spacecraft system. The net result is a system attractive for overall practical application. The liquid ammonia system has been initially introduced into the Application Technology Satellite Project as a flight experiment.

#### Advanced Technology Satellite

The Application Technology Satellite Project is the National Aeronautics and Space Administration's means of conducting development and flight testing of promising new technology that is fundamental to a number of satellite applications. Two such areas of immediate interest are passive gravity gradient attitude control systems, and the technology associated with the stationary orbit. The ATS multiple mission spacecraft provides a relatively large, adaptable payload capability designed to achieve long life in circular medium-altitude orbits or the synchronous, equatorial orbit. Gravity gradient stabilization provides vehicle orientation in the medium altitude orbit; either gravity gradient or spin stabilization, depending on mission objectives, is used for vehicle orientation in the synchronous orbit.

A fast heat-up liquid ammonia resistojet experiment was integrated, during the very late stages of spacecraft operations, into the ATS-1 spacecraft

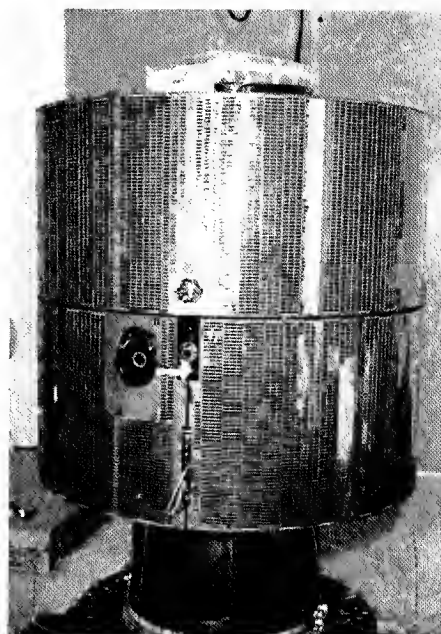


FIGURE 4 - ATS-1 SPACECRAFT

shown in Figure 4, a synchronous orbit spin stabilized vehicle. The experiment system was initially developed by the Avco Corporation under contract to Lewis Research Center and a model of the system was integrated by Goddard Space Flight Center and Hughes Aircraft Company into the spacecraft. The prime objective of the experiment was to determine the magnitude of the effective thrust output of each of two thrusters by measuring spin period change after thruster firings.

#### System Description

The system is shown schematically in Figure 5 and pictorially in Figure 6.

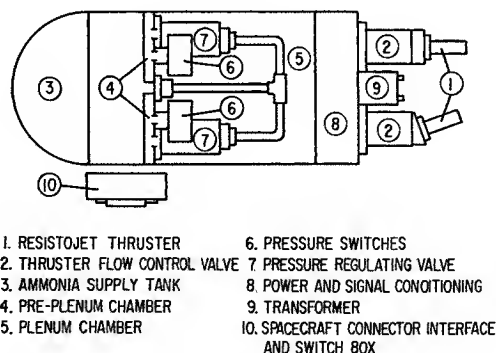


FIGURE 5 - SCHEMATIC OF RESISTOJET SYSTEM

The system is comprised of the following sub-systems: (a) two thrusters assemblies, each consisting of a fast heat thruster (item 1) and thruster flow control solenoid valve (item 2); (b) liquid ammonia supply tank (item 3); (c) propellant feed system, consisting of a pre-plenum (item 4), plenum (item 5), pressure switches (item 6), and pressure regulating valves (item 7); (d) power and signal conditioning (items 8 and 9); and (e) spacecraft

connector - interface and switch box (item 10).

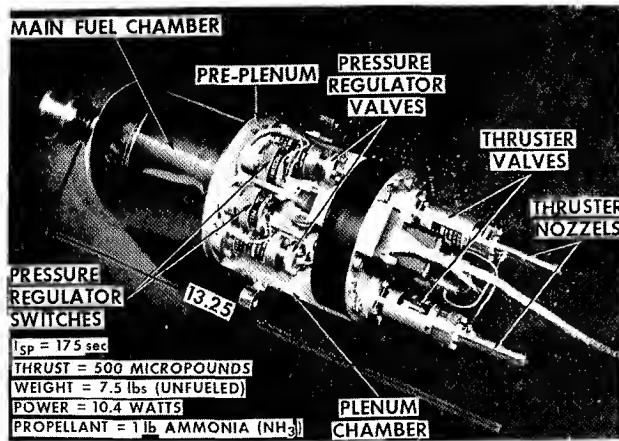


FIGURE 6 - AVCO AMMONIA RESISTOJET SYSTEM

#### Resistojet Thruster

The basic element of the fast heat-up thruster is a single-pass heat exchange tube connected to an exit nozzle. The heater tube is surrounded by a support tube. High current electrical connections are located on the support tube and mounting base of the heater tube; the outer support tube is insulated from the base except through the heater tube. The ammonia propellant is heated as it passes through the heater tube and the thermal energy is converted to kinetic energy during expansion through the exit nozzle. The mean specific impulse is about 175 seconds at 7 watts input power. A diagram of a stainless steel heater-nozzle element is shown in Figure 7.

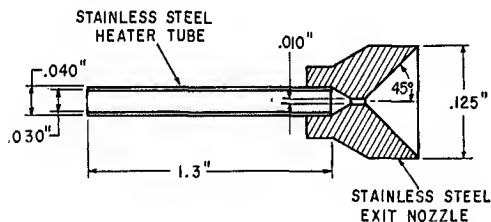


FIGURE 7 - STAINLESS STEEL HEATER-NOZZLE ELEMENT

Average thrust output of the thrusters was designed to be 500 micropounds; the thrust level was selected based on requirements for inversion maneuvers for gravity gradient spacecraft. Subsequent ground tests after integration of the experiment revealed, however, that because of recessing of the thrusters in the spacecraft as shown in Figure 8 and the presence of a protective teflon shield around the thruster area, a severe reduction in effective thrust level was experienced. The net effective thrust level of one thruster was zero; the other thruster was about 300 micropounds rather than 500 micropounds, the original design level.

#### Feed System

A schematic of the propellant feed system is shown in Figure 9. The overall function of the feed system is to provide gaseous ammonia to the thrusters at the required plenum chamber pressure

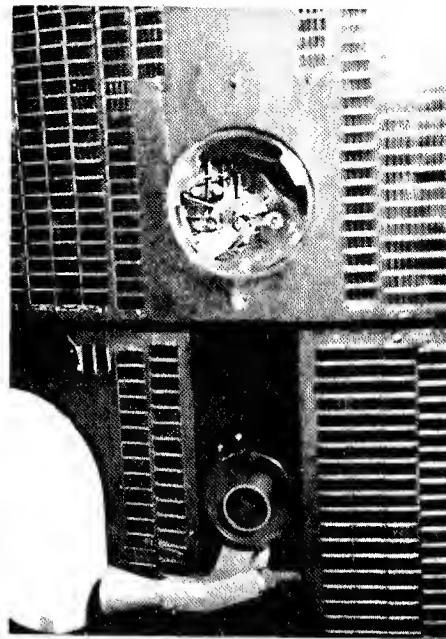


FIGURE 8 - SPACECRAFT SHOWING RESISTOJET ENGINES

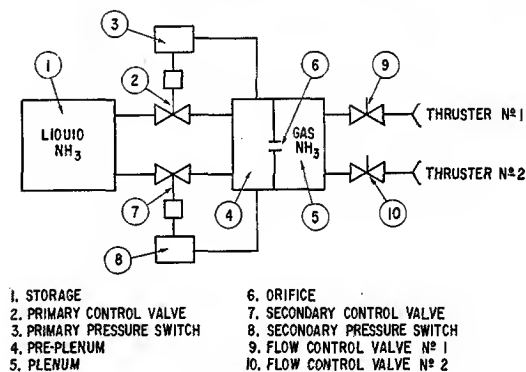


FIGURE 9 - SCHEMATIC OF PROPELLANT FEED SYSTEM

of approximately 6 psia  $\pm$  10%. The elements of the feed system are the supply tank, pre-plenum, a plenum chamber and a pressure regulator that consists of a solenoid valve activated by a pressure switch. The pressure switch senses pressure in the pre-plenum chamber. The use of a small sensing volume makes it possible to hold the plenum pressure to within limits as low as 1 percent with either gas or liquid flow through the pressure regulating valve. The basic tradeoff is between valve cycling rate and pressure regulation. Better pressure regulation requires higher valve cycling rates.

During system operation, as ammonia vapor is withdrawn through one of the thruster flow control valves, the pressure in the plenum and pre-plenum decreases. As the pressure drops below the pressure switch close limit, the switch closes in electrical circuit which opens a normally closed pressure regulating solenoid valve allowing ammonia to enter the pre-plenum from the supply tank. The pre-plenum pressure then increases to the open limit of the pressure switch; the switch opens the electrical circuit to the valve and the valve closes stopping the flow from the supply. This sequence automatically repeats as a function of the withdrawal rate



of the propellant from the plenum chamber.

The feed system for the experiment was designed to include a redundant pressure switch-regulation valve element. The redundant switch has a pressure deadband which encompasses that of the primary switch. Should a pressure regulating valve leak, it is possible for the pressure in the plenum chamber to rise to the vapor pressure of the ammonia in the supply tank. If the pressure exceeded 100 psia the more sensitive primary pressure switch might damage resulting in a permanent no flow, open signal from the switch. To insure system operation in the event of this type of failure, the secondary pressure switch has a significantly higher range (to 300 psia) but consequently is less sensitive and has a wider dead band. The pressure switch settings and regulation system characteristics are described in Table 2.

TABLE 2

Pressure Switch Settings

Primary Switch	Closes (valve power on)	5.60 psia
	Open (valve power off)	6.10 psia
Secondary	Closes (valve power on)	4.50 psia
	Open (valve power off)	7.10 psia

Regulation System Characteristics

Gas Feed - Pressure Regulation Valve  
Cycle rate - 35-55 cycles/minute  
Maximum Plenum Pressure Variation  $\pm .9\%$   
Mass Flow -  $3.5 \times 10^{-6}$  lbs/second

Command and Telemetry

Command and telemetry for the system is shown in Table 3.

TABLE 3

Command

1. Power Regulator ON
2. Power Regulator OFF
3. Thruster No. 1 ON
4. Thruster No. 1 OFF
5. Thruster No. 2 ON
6. Thruster No. 2 OFF

Telemetry

1. Primary Regulating Valve Voltage
2. Secondary Regulating Valve Voltage
3. Plenum Pressure
4. Input Heater Current
5. Thruster #1 Valve Voltage
6. Thruster #2 Valve Voltage
7. Thruster #1 Heater Voltage
8. Thruster #2 Heater Voltage

The system was designed to operate from a negative 24 volt direct current power. Electrical circuits for the resistojet heater tubes were designed to convert the direct current input to alternating current. As shown in Table 3 the command capability was limited. Once the power regulator command was given, the system could only be turned on or off with a thruster firing, and all components were activated simultaneously. Consequently, telemetry circuits were activated only during thruster firing.

Experiment Results

The resistojet engines were fired for one hour each during the first experiment operation. Data analysis disclosed that thruster firing resulted in no change in spacecraft spin period; however, the first interpretation of the system telemetry indicated that the system was operating normally. Data from telemetry was similar to data acquired in a final system test during spacecraft thermal vacuum testing.

An investigation of the conflicting results was initiated and cannot be covered in complete detail in this paper but only summarized. The original effect of thrust reducing due to interfaces with the spacecraft was surmised to be more severe than expected but could not conclusively be established as a mechanism of failure in the experiment.

An analysis of the flight telemetry data, test data, system functioning, and history of the system prior to launch was conducted. Conclusions are that the system was launched with an empty or near empty supply tank. A rework of the system during integration resulted in damage to the plenum pressure transducer resulting in a major leak causing a high mass flow to occur during system firing. An inadvertent firing in the thermal vacuum testing of the spacecraft (a firing which, under normal operation would not be catastrophic) occurred resulting in near depletion of the propellant supply. Lack of comprehensive system characterization and the limited command and telemetry diagnostic capability precluded the identification of the major leak during spacecraft ground tests.

ATS Follow-on Experiment

A follow-on experiment to be conducted on the next ATS spin stabilized spacecraft retains the basic thruster and feed system concepts. This experiment, however, incorporates engines with two different thrust levels. Furthermore, significant design changes have been made in configuration, command capability, and telemetry.

One thruster incorporates a porous plug to reduce its thrust level to 10 micropounds, a value suited for gravity gradient stabilized spacecraft. The other thruster, which also has a porous plug, will operate at approximately 10 micropound thrust level.

The command and telemetry functions are shown in Table 4.

TABLE 4

Command

1. Power Regulator ON
2. Power Regulator OFF
3. Thruster No. 1 ON/OFF
4. Thruster No. 2 ON/OFF
5. Thruster Heat Enable
6. Pressure Regulation System ON

Telemetry

1. Regulator ON/OFF
2. Primary Regulation Valve Current
3. Secondary Regulation Valve Current
4. Plenum Pressure

5. Thruster No. 1 Heater Current
6. Thruster No. 2 Heater Current
7. Thruster No. 1 or 2 Heater Current
8. Flow Valve No. 1 or 2 Current
9. System Temperature

The configuration allows the thrusters to protrude approximately one-half inch beyond the spacecraft outer perimeter, thus eliminating the previous problem of thrust reduction. In addition, examination of the command modes, utilizing the same number of commands, shows a significant increase in ground test and diagnostic capability of the system. For example the thrusters can be fired either hot or cold by use of the Thrust Head Enable command; cold firing allows testing of the system in the atmosphere. Either thruster can be fired to deplete the plenum chamber; thus, the dynamic response of the plenum pressure transducer can be observed provided the Pressure Regulation System ON command is not activated. These modes of operation allow assessment of the feed system operability at any time during the pre-launch period. The telemetry system is activated upon initial application of power providing data during non firing times as well as when the thrusters are being fired. The system is shown in Figure 10.

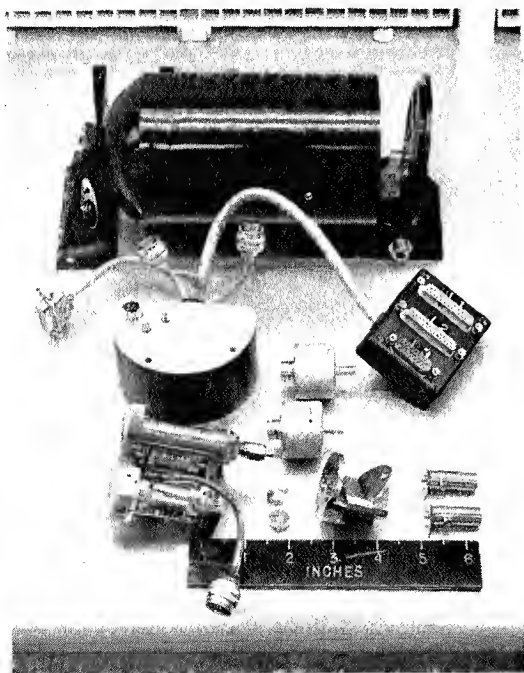


FIGURE 10 - ATS 10 MICROPOUND/100 MICROPOUND EXPERIMENT PACKAGE

The system was selected for application for station keeping on the ATS-D and ATS-E gravity gradient synchronous orbit missions. The objective of the technology experiment is to assess the total system performance and characteristics relevant to its prime function for station keeping in the ATS-D and ATS-E missions.

#### Advanced Thermal Storage Thrusters

In the thermal storage thruster power is continuously supplied to the heater element and normally only the propellant flow is pulsed. The heat capacity of the thermal storage unit is sufficiently great that the temperature of the heater element remains essentially constant during short propellant

pulses. The principal advantage of the thermal storage thruster is the achievement of high  $I_{sp}$  for a low peak power during pulsed mode operations. The heater elements in these units are designed to raise the gas temperature to 2000°F. To minimize the required power and maintain high propulsion efficiency, the thrusters employ thermal insulation techniques.

#### Single Jet Thermal Storage Thruster

Figure 11 shows a sectioned model of a single jet thruster developed by General Electric Company under contract to Goddard Space Flight Center depicting its physical design characteristics.

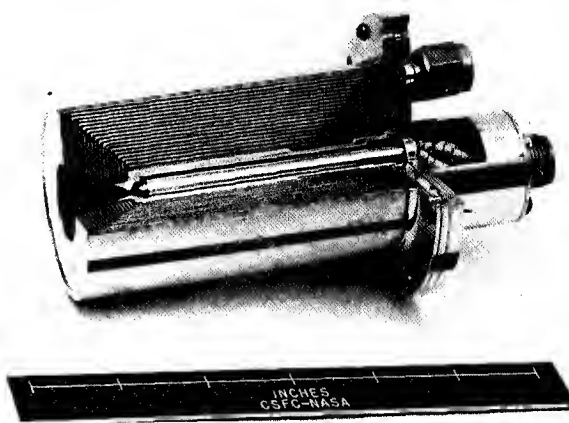


FIGURE 11 - G. E. SINGLE JET THERMAL STORAGE THRUSTER

#### Description

The external envelope of the thruster is 2 inches in diameter by 5 inches long with a circular mounting flange. The thruster design is unique not only because of the heat shield technique but also because the thruster employs a modular design.

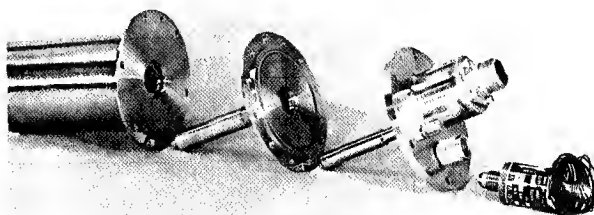


FIGURE 12 - DISASSEMBLED SINGLE JET THRUSTER

As shown in Figure 12 the thruster consists of three modular components: a thruster body; a heat shield package; and an electrical heater unit. The design concepts permits interchangeability of the

three precision made modules. Assembly or disassembly is simple but reliable using bolt screw techniques; especially significant is that the integrity of the hardware is maintained even if many disassembly-assembly operations are necessary. This important design feature has proven its merit in many other types of space flight hardware and is a significant advantage in electrothermal thruster design. Design changes altering thruster performance characteristics can be made readily without affecting the design interfaces or envelope, thus eliminating the need for major redesign efforts and extensive requalification programs. Damage incurred by a thruster would require simply the repair or replacement of one module and rarely would result in total destruction of a thruster, common to many aerospace designs.

The basic assembly is depicted as three concentric cylindrical modules components with mating common flange interfaces. The heat shield package is a series of forty concentric nichrome shells (0.003 inches wall thickness) separated by wraps of 0.014 inch diameter nichrome wire. The heater element is a standard calrod type swaged heater consisting of a magnesia mandrel wrapped with Platinum wire, a Hastelloy X sheath, and pneumatically impacted boron nitride fill. The thruster body is a Hastelloy X shell containing a sonic nozzle. The heater element fits into the thruster body forming an annular propellant path. The heat shield package fits over the thruster body. In addition, a thermocouple is built into the calrod heater element.

#### Performance

Performance properties are determined by direct measurements of thrust, thrust chamber pressure, chamber and body temperatures, and propellant flow rate.

The thruster was designed to produce twenty millipounds of thrust at twenty-two pounds per square inch chamber pressure. Table 5 provides the pertinent performance data of the single jet thruster.

TABLE 5

Weight	- 1.75 lbs.
Thrust	- 20 millipounds
Chamber Pressure	- 22.5 psia
Propellant	- ammonia
Power	- 30 watt    33 watt    37 watt
Core Temperature (no flow)	- 1840°F    2000°F    2100°F
Specific Impulse	- 228 sec    254 sec    266 sec
Heat up Procedure to Stabilize at 1840°F (no flow)	
5 watts for 1 hour	
30 watts for 4 hours	
Equilibrium temperature (steady flow thrusting)	
510°F after 4 hours	
Initial power - 30 watts	current-limited to
Final power - 22 watts	2.2 amps voltage
	variable

Figure 13 shows a curve of laboratory measurements of power versus temperature of the single jet thruster. Figure 14 shows specific impulse versus temperature. The resultant tests and examination of the design indicates that improved power performance can be achieved through some mechanical

redesign without affecting the integrity of the modular concept. Primary heat loss is by means of direct thermal conduction from the heat element to the rear flange. Insulation of the heater element from the flange could reduce power requirements 30 percent.

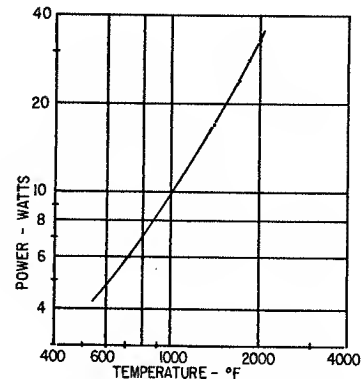


FIGURE 13 - POWER VERSUS TEMPERATURE TSK-2000-1P

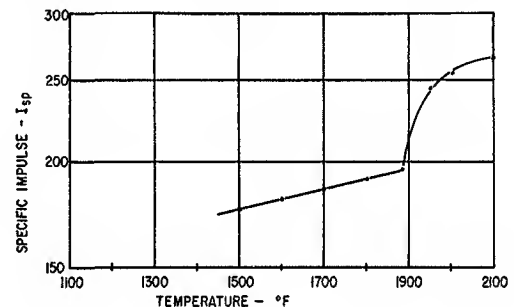


FIGURE 14 - SPECIFIC IMPULSE VERSUS TEMPERATURE TSK-2000-1P

#### Multijet Thruster

Figure 15 shows a sectioned model of a multijet thruster depicting the physical design characteristics.

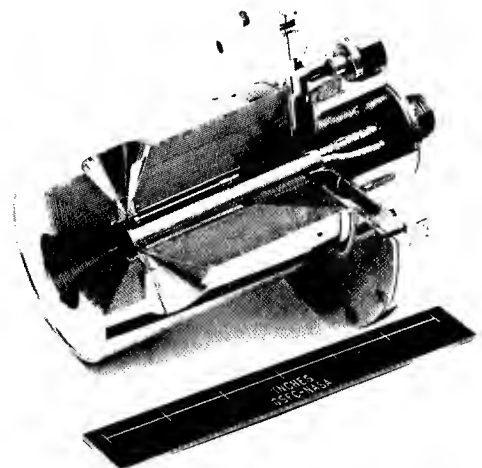


FIGURE 15 - G. E. MULTIJET THERMAL STORAGE THRUSTER

The unit is essentially one thruster with five nozzles - one in the axial direction and one in



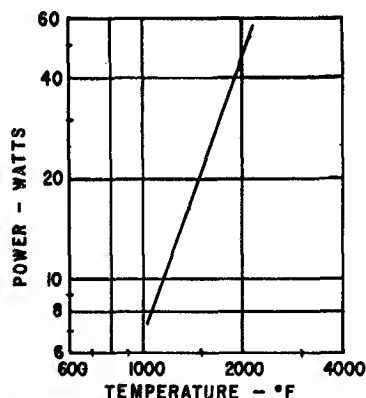
each of four radial directions 90° apart. The multijet thruster concept is being considered for application to combined attitude control-station keeping subsystems. The photograph of Figure 15 is a model of one type of thermal storage multijet currently under development by General Electric Company to Goddard Space Flight Center. It is approximately 5 inches long, two inches in diameter, and produces thrust levels of from 5 to 50 millipounds. Its construction is similar to the single jet; however, it is a two module unit consisting of the heater element module and the thruster body - heat shielding module.

Table 6 provides the pertinent performance characteristics of the multijet thruster.

TABLE 6

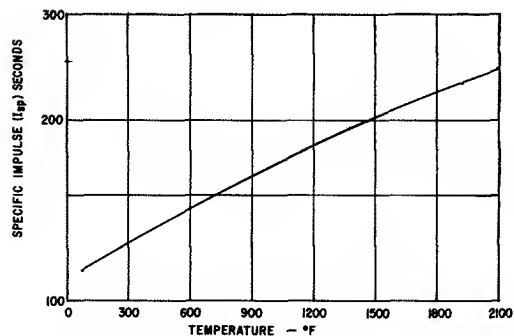
Weight - 2.5 lbs.			
Thrust - end nozzle 50 millipounds			
radial nozzle - 20 millipounds			
Chamber pressure - 22.5 psia			
Propellant - ammonia			
Power - 16 watts	43 watts	50 watts	
Core Temperature - 1378°F	1925°F	2085°F	
Specific Impulse - 195 sec.	230 sec.	244 sec.	

Figure 16 shows power versus temperature, and Figure 17 shows specific impulse versus temperature of the thermal storage multijet thruster.



POWER VERSUS TEMPERATURE, END NOZZLE  
TMJ-2000-5P

FIGURE 16 - POWER VS. TEMPERATURE OF  
MULTIJET THRUSTER



SPECIFIC IMPULSE VERSUS TEMPERATURE, END NOZZLE - TMJ-2000-5P

FIGURE 17 -  $I_{sp}$  VS. TEMPERATURE OF MULTIJET THRUSTER

A critical problem area requiring further development effort is improvement in the design of calrod type heater elements for reliability. Increased steady state heater life is desirable; heaters which can operate continuously for more than two to three years. In addition it is desirable for the heaters to be able to be cycled from ambient temperature to 2000°F for more than 1000 cycles. Neither objective has yet been achieved. Development efforts in this area will be continued.

#### Radioisotope Thruster

Although not an electrothermal system, a thruster called a radioisotrojet has also been developed by General Electric under contract to Goddard Space Flight Center. The thruster, a model of which is shown in Figure 18, utilizes an isotope capsule as a heat energy source in place of a heater element. The thruster was developed and tested under a joint NASA/AEC coordinated program. The design is founded on original efforts conducted on the electrothermal storage thrusters and presents a potential solution for extreme long life applications. This thruster is discussed in detail in a paper titled "Design and Demonstration of a Radioisotrojet" by R. Viventi and W. Isley.

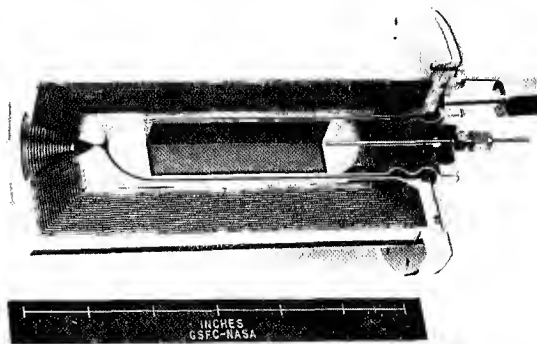


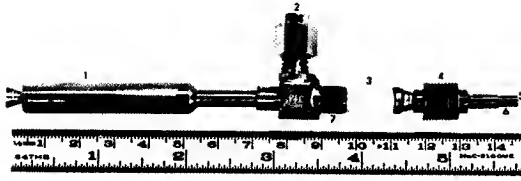
FIGURE 18 - RADIOISOTOPE THRUSTER

#### Concentric - Tubular Regenerative Thruster\*

Electrothermal thrusters discussed thus far operate at power levels under 50 watts. For applications where power requirements are not restrained severely, Marquardt is currently developing a 10 millipound thrust level resistojet, shown in Figure 19, under NASA sponsorship. The thruster can be employed for attitude control and station keeping of manned or large unmanned spacecraft, or orbit changing maneuvers such as for lifting a satellite to synchronous altitude. This thruster is a concentric - tubular, regenerative design fabricated from vapor-deposited rhenium. The rhenium tubes are fused into a one-piece, series - connected structure by electron beam welding. A propellant temperature of approximately 4000°F is achieved which corresponds to a specific impulse of 360 and 740 seconds with ammonia and hydrogen propellant respectively. The engine incorporates a ceramic-to-metal seal for electrical insulation and operates at 6 volts and 40 amps. The unit is currently undergoing development testing.

\*Courtesy of Marquardt Corporation

MODEL I 10-MILLIPOUND RESISTOJET  
FULL SIZE



1. 10 mlb RESISTOJET
2. PROPELLANT CONNECTOR
3. ELECTRIC INSULATOR SEAL
4. THERMAL EXPANSION BELLOWS
5. VACUUM JACKET VENT
6. (+) TERMINAL
7. (-) TERMINAL

FIGURE 19 - MARQUARDT RESISTOJET

Conclusions

As stated previously, the electrothermal systems and thrusters discussed do not encompass the status of all efforts being conducted in electrothermal system technology because of classification or proprietary interests. As examples, however, they do illustrate the trend toward the application of the electrothermal system to advanced spacecraft applications as controls and station keeping subsystems. The problems with which designers are now confronted are no longer limited to the propulsion system as an entity but as a system totally integrable with a spacecraft and its mission objectives. The technological status of electrothermal systems is not to a point of completion; hardware is not truly "off the shelf" and that which is considered standard has yet to prove itself by frequent successful application. It is to these ends that research, development, and application efforts should be directed.

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